COMPARATIVE COST ASSESSMENT OF PLANETARY MISSIONS

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FOREWORD

The analysis and results contained herein were performed by the Advanced Planning and Analysis Division of Science Applications, Inc. during the Summer of 1981. Key analysts for this study were Messrs. Daniel Spadoni (costing) and John Niehoff (mission analysis). The results were submitted to Mr. Ken Russ, the technical monitor, earlier for incorporation into planning activities relevant to NASA's SSEC Summer Study. They are documented here for formal presentation as needed.

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The purpose of this study is to explore the cost differences resulting from implementing a series of representative solar system exploration missions in either ballistic or low-thrust flight modes. Of particular interest are cost comparisons of missions using a Solar Electric Propulsion Stage (SEPS) delivery system with ballistic equivalent mission designs.

The scope of the study is defined by the mission set, delivery modes, and cost elements to be studied. The mission set consists of six missions including two asteroid missions, a comet mission, a Mercury mission, and two outer planet missions. The delivery modes include SEPS and two Ballistic options: 1) a baseline option using traditional ballistic mission design guidelines, and 2) a parity option which forces the ballistic design to match the operational flexibilities of the SEPS option. The costing analysis includes consideration of both Project costs and Transportation costs. Project costs consist of spacecraft development and flight operations costs and also include the cost of all spacecraft propulsion. Transportation costs include the cost of launch and earth escape. They also include the cost of SEPS procurement and operations (even though on some missions the SEPS is retained through the mission encounter phase). Further definitions of these elements of the study scope are included below in the Scope and Assumptions section.

INTRODUCTION

- PURPOSE

  TO INVESTIGATE A SERIES OF REPRESENTATIVE SOLAR SYSTEM MISSIONS WITH THE INTENT OF DETERMINING THE COST DIFFERENCES RESULTING FROM IMPLEMENTING THESE MISSIONS IN EITHER THE BALLISTIC OR LOW-THRUST FLIGHT MODES

- SCOPE

  - MISSION SET
    - THREE PLANETARY MISSIONS
    - THREE PRIMITIVE BODY (Comets/Asteroids) MISSIONS
  - FLIGHT MODES
    - BALLISTIC (including planetary swingbys)
    - LOW-THRUST (utilizing the Solar Electric Propulsion Stage Concept)
  - COST CATEGORIES
    - PROJECT COSTS (Development and Operations)
    - TRANSPORTATION COSTS (Earth launch/escape and Interplanetary Low-Thrust)

The table on the facing page summarizes the eighteen mission options analyzed for this study. The six mission cases selected for detailed study are:

1. Near-Earth Asteriod Rendezvous (with follow-on targets in the main belt)
2. Comet Tempel-2 Rendezvous
3. Main-Belt Asteriod Rendezvous
4. Mercury Elliptical Orbiter
5. Outer Planet (Uranus/Neptune) Dual Probe Flyby
6. Saturn Dual Probe Orbiter (Saturn and Titan probes)

The principal mission parameters affecting the cost assessment analysis are shown for each option and include such cost drivers as shuttle upper stage utilization, spacecraft mass and propulsion, flight mode and mission duration times. The latter of these parameters, which mainly affect the Project category costs, are discussed in the individual mission sections.

The shuttle upper stage utilization, which affects the Transportation category costs, is summarized in the following table, showing the number of units required for each delivery option.

	SEPS	Ballistic Baseline	Ballistic Parity
Shuttle	6	6	12
IUS(II)	4	1	0
Centaur(WB)	2	5	10
Star 48	0	2	0
SEP Stage	6	0	0

MISSION SET DEFINITION SUMMARY

DELIVERY OPTIONS	MISSION PARAMETERS	MISSION SET: 1. NEAR-EARTH ASTEROID RENDEZVOUS	2. COMET TEMPEL-2 RENDEZVOUS	3. MAIN-BELT ASTEROID RENDEZVOUS	4. MERCURY ELLIPTICAL ORBITER	5. OUTER PLANET DUAL PROBE FLYBY	6. SATURN DUAL PROBE ORBITER
SEPS BASELINE	TARGETS	3 ASTEROIDS	TEMPEL-2	4 ASTEROIDS	MERCURY	URANUS & NEPTUNE	SATURN & TITAN
	SHUTTLE UPPER STAGES	IUS(II)	IUS(II)	IUS(II)	IUS(II)	CENTAUR(WB)	CENTAUR(WB)
	NET S/C MASS	300 KG	640 KG	605 KG	525 KG	725 KG	760 KG
	PROBE MASSES	-	-	-	-	275 KG/PROBE (2)	250(TITAN), 300(SATURN)
	S/C PROPULSION	NONE	NONE	NONE	SOLID/HYDRAZINE	HYDRAZINE	EARTH-STORABLE
	LAUNCH DATE	DEC 1987	JUN 1990	OCT 1988	ONCE/YEAR (OR MORE)	ONCE/YEAR (1985-95)	ONCE/YEAR
	FLIGHT MODE	MULTI-REV	DIRECT	MULTI-REV	MULTI-REV	SEEGA SWINGBY	SEEGA
	PLANET SWINGBYS	NONE	NONE	NONE	NONE	EARTH, URANUS	EARTH
	TRIP TIME	5.2 YEARS	3.6 YEARS	7.9 YEARS	1.7 YEARS	10.2 YEARS	5.4 YEARS
	STAY TIME	2 MONTHS/TARGET	12 MONTHS	2 MONTHS/TARGET	6 MONTHS	-	24 MONTHS
BALLISTIC BASELINE	TARGETS	2 ASTEROIDS	TEMPEL-2	2 ASTEROIDS	MERCURY	URANUS & NEPTUNE	SATURN & TITAN
	SHUTTLE UPPER STAGES	CENTAUR(WB)	CENTAUR(WB)	CENTAUR(WB)	IUS(II)/STAR 48	CENTAUR(WB)/STAR 48	CENTAUR(WB)
	NET S/C MASS	550 KG	755 KG	730 KG	555 KG	725 KG	760 KG
	PROBE MASSES	-	-	-	-	275 KG/PROBE (2)	250(TITAN), 300(SATURN)
	S/C PROPULSION	SPACE-STORABLE(2 STG)	EARTH-STORABLE(2 STG)	EARTH-STORABLE(2 STG)	EARTH-STORABLE	HYDRAZINE	EARTH-STORABLE
	LAUNCH DATE	OCT 1988	JUL 1989	JUL 1988	JUL 1994	DEC 1992	JUL 1998
	FLIGHT MODE	MULTI-IMPULSE DIRECT	MULTI-IMPULSE DIRECT	MULTI-IMPULSE SWINGBY	DOUBLE VENUS SWINGBY	MULTI-PLANET SWINGBY	SWINGBY
	PLANET SWINGBYS	NONE	NONE	MARS	VENUS, VENUS	JUPITER, URANUS	JUPITER
	TRIP TIME	4.5 YEARS	4.5 YEARS	4.1 YEARS	2.4 YEARS	8.2 YEARS	4.8 YEARS
	STAY TIME	$\geq$ 2 MONTHS/TARGET	12 MONTHS	$\geq$ 2 MONTHS/TARGET	6 MONTHS	-	24 MONTHS
BALLISTIC PARITY	TARGETS	3 ASTEROIDS	TEMPEL-2	4 ASTEROIDS	MERCURY	URANUS & NEPTUNE	SATURN & TITAN
	SHUTTLE UPPER STAGES	3 CENTAURS W/OOA[1]	CENTAUR(WB)	3 CENTAURS W/OOA[1]	CENTAUR(WB)	CENTAUR(WB)	CENTAUR(WB)
	NET S/C MASS	550 KG	755 KG	730 KG	555 KG	725 KG	760 KG
	PROBE MASSES	-	-	-	-	275 KG/PROBE (2)	250(TITAN), 300(SATURN)
	S/C PROPULSION	SPACE-STORABLE(3 STG)	SPACE-STORABLE(2 STG)	SPACE-STORABLE(3 STG)	EARTH-STORABLE	EARTH-STORABLE	EARTH-STORABLE
	LAUNCH DATE	OCT 1988	AUG 1990	JUL 1988	ONCE EVERY 2-3 YRS	ONCE/YEAR (1985-95)	ONCE/YEAR
	FLIGHT MODE	MULTI-IMPULSE DIRECT	MULTI-IMPULSE DIRECT	MULTI-IMPULSE SWINGBY	MULTI-VENUS SWINGBY	ΔVEGA (3+) SWINGBY	ΔVEGA (3+)
	PLANET SWINGBYS	NONE	NONE	MARS	VENUS (1-3 TIMES)	EARTH, URANUS	EARTH
	TRIP TIME	5.9 YEARS	3.5 YEARS	9.9 YEARS	$\leq$ 3.4 YEARS	14.8 YEARS	7.0 YEARS
	STAY TIME	$\geq$ 2 MONTHS/TARGET	12 MONTHS	$\geq$ 2 MONTHS/TARGET	6 MONTHS	-	24 MONTHS

1. OOA: ON-ORBIT-ASSEMBLY

The table on the facing page summarizes the costs estimated for each of the mission options analyzed for this study. All costs are presented in FY '82 million dollars.

Total costs are shown separated into two categories. The Project category contains all costs associated with mission hardware development and mission operations. The Transportation category includes all costs associated with the launch system and with solar electric propulsion.

In general, the SEPS Options have the lowest Project costs but the highest Transportation costs. The exceptions to this are, in the Project Category, the two outer planet missions which require a large retro propulsion system despite the low-thrust flight mode, and in the Transportation Category, the Ballistic Parity asteroid missions which require multiple launches.

The Total costs, viewed across the set of 18 mission options, do not present a consistent cost trend favoring any one particular option. In only one case, the Near-Earth Asteroid Rendezvous, the SEPS Option shows significant cost savings potential over the ballistic options. However, this is due primarily to the unique mission payload assumptions used for this particular mission. In two cases, the Comet Tempel-2 Rendezvous and the Mercury Elliptical Orbiter, the cost estimates of the three options are within 10% of each other, and for the Saturn Dual Probe Orbiter the costs are nearly within 10%. Therefore, any conclusions regarding the benefits of any one delivery mode should not be made strictly on the basis of cost.

COST* COMPARISONS OF DELIVERY MODE OPTIONS

COST CATEGORY	DELIVERY OPTION	MISSION SET (6 CASES)					
		1. NEAR-EARTH ASTEROID RENDEZVOUS	2. COMET TEMPEL-2 RENDEZVOUS	3. MAIN-BELT ASTEROID RENDEZVOUS	4. MERCURY ELLIPTICAL ORBITER	5. OUTER PLANET DUAL PROBE FLYBY	6. SATURN DUAL PROBE ORBITER
PROJECT	SEPS	127.8	355.3	352.8	198.0	622.1	873.7
	B.BASE	311.5	434.0	420.2	259.2	600.1	848.2
	B.PARITY	385.4	460.0	607.6	257.2	731.3	889.1
TRANSPORTATION	SEPS	161.7	159.6	171.0	148.7	175.2	177.9
	B.BASE	98.0	98.0	98.0	79.0	104.0	98.0
	B.PARITY	367.0	98.0	367.0	98.0	98.0	98.0
TOTAL	SEPS	289.5	514.9	523.8	346.7	797.3	1051.6
	B.BASE	409.5	532.0	518.2	338.2	704.1	946.2
	B.PARITY	752.4	558.0	974.6	355.2	829.3	987.1

* FY'82 Millions of Dollars

In order to provide some insight into the cost benefits/penalties of any one of the delivery modes implemented at a program level rather than a mission level the individual mission cost estimates presented on the previous page have been summed as a mission set below. Note that the Near-Earth Asteroid Rendezvous mission has not been included in these totals. The reason being that it is unlikely that two asteroid rendezvous missions would be included in a program. Also, costs comparisons would otherwise become biased in favor of SEPS which has a distinct advantage for asteroid rendezvous missions. The remaining five missions are considered to be a mixed, representative set of new planetary mission initiatives.

The SEPS option requires the lowest Project costs and highest Transportation costs; total costs are intermediate between the two Ballistic Options. The Ballistic Baseline option has an intermediate Project cost and the lowest Transportation and Total costs. The Ballistic Parity option produces the highest Project cost, an intermediate Transportation cost, and the highest Total cost.

The lowest Total cost of the Ballistic Baseline is achieved by making concessions in either launch flexibility or mission scope, both of which can be interpreted as constraints on exploration objectives. Both SEPS and Ballistic Parity options remove these constraints, but SEPS does so at much less cost. Furthermore, the SEPS option maintains good flight time performance and avoids the questionable requirement of on-orbit upper stage assembly. The design feasibility of very large spacecraft retro systems is also avoided.

Two conclusions are apparent from these results. They are as follows:

1) Proceeding with solar system exploration within a "ballistic baseline" implementation philosophy will incur constraints in both planning and achievements;

2) Implementing SEPS can avoid these limitations at low additional cost when considered on an overall program basis.

CUMULATIVE[1] COST COMPARISONS

COST CATEGORY	COMPARATIVE PARAMETER	DELIVERY OPTION		
		SEPS	BALLISTIC BASELINE	BALLISTIC PARITY
PROJECT	COST[2]	$2401.9M	$2561.7M	$2945.2M
	VARIANCE[3]	-6.2%	–	13.0%
TRANSPORTATION	COST	832.4M	477.0M	759.0M
	VARIANCE	74.5%	–	59.1%
TOTAL	COST	$3234.3M	$3038.7M	$3704.2M
	VARIANCE	6.4%	–	21.9%

1. EXCLUDING NEAR-EARTH ASTEROID RENDEZVOUS MISSION
2. FIXED FY'82 DOLLARS
3. RELATIVE TO BALLISTIC BASELINE OPTION

The six missions selected for detailed analysis are representative of a broad scope in solar system exploration. Included are two asteroid missions, a short-period comet mission, an inner planet mission and two outer planet missions. The mission objectives encompass the spectrum from initial exploration to intense investigation, and thus are typical of the kinds of missions under consideration for the 1985 to 2000 time frame.

A characteristic common to the six missions is a requirement for complex flight profiles. Such flight mode complexities arise from a variety of reasons such as multi-revolution low-thrust trajectories, single or multiple gravity-assist planet swingbys, and multiple deep space ΔV maneuvers. In turn, these flight mode requirements are dictated by a variety of mission/performance objectives such as multiple targets and tradeoffs between payload mass delivery, trip time and launch energy. Details of the various flight profiles are presented in the individual mission sections.

MISSION SET

MISSION	FLIGHT MODE: SEPS OPTION	FLIGHT MODE: BALLISTIC BASE OPTION	FLIGHT MODE: BALLISTIC PARITY OPTION
1. NEAR-EARTH ASTEROID RENDEZVOUS	MULTI-REV	MULTI-IMPULSE DIRECT	MULTI-IMPULSE DIRECT
2. COMET TEMPEL-2 RENDEZVOUS	DIRECT	MULTI-IMPULSE DIRECT	MULTI-IMPULSE DIRECT
3. MAIN-BELT ASTEROID RENDEZVOUS	MULTI-REV	MULTI-IMPULSE SWINGBY[1]	MULTI-IMPULSE SWINGBY[1]
4. MERCURY ELLIPTICAL ORBITER	MULTI-REV	DOUBLE VENUS SWINGBY	MULTI-VENUS SWINGBY
5. OUTER PLANET DUAL PROBE FLYBY	SEEGA SWINGBY[2]	MULTI-PLANET SWINGBY[3]	ΔVEGA SWINGBY[2]
6. SATURN DUAL PROBE ORBITER	SEEGA	JUPITER SWINGBY	ΔVEGA

1. MARS
2. URANUS
3. JUPITER AND URANUS

The main parameter of the study driving the trade-off analysis is flight mode implementation, as represented through the choice of propulsion systems selected for performing the mission.

Propulsion system options are classified by mission phase: launch and Earth escape, interplanetary flight, and target encounter. Launch and escape systems are limited to those systems which are most likely to be available for use in solar system exploration. Specifically, the Space Transportation System (Shuttle) is the primary launch vehicle to Earth orbit and the available upper stages for interplanetary injection are the Inertial Upper Stage (II) and the Wide-Body Centaur. In addition, the Star 48 solid motor is assumed to be available as a final kick stage if needed.

Options available for both the interplanetary and encounter phases are either a solar electric propulsion stage or a chemical propulsion system. For most mission options, the same propulsion system is assumed to perform both interplanetary and encounter maneuvers. However, some low-thrust mission options require both a SEP stage and a chemical system.

PROPULSION OPTIONS

- LAUNCH AND ESCAPE
  - o SHUTTLE LAUNCH SYSTEM
  - o UPPER STAGES
    - \- IUS(II)
    - \- CENTAUR (WIDE BODY)
  - o STAR 48 KICK STAGE
- INTERPLANETARY FLIGHT
  - o SOLAR ELECTRIC PROPULSION STAGE (SEPS)
  - o BALLISTIC (baseline and SEPS "parity")
- ENCOUNTER
  - o HYDRAZINE MONOPROP
  - o SOLID MOTOR
  - o EARTH-STORABLE LIQUID (1 and 2 stages)
  - o SPACE-STORABLE LIQUID (2 and 3 stages)
  - o SEPS (comet and asteroid rendezvous)

The following general guidelines and assumptions were used in defining and analyzing the mission options.

The time frame of mission launch opportunities, 1985 to 2000, is consistent with current NASA long-range planning periods.

Mission flight hardware, for the most part, is developed through maximum use of state-of-the-art technology. This involves a high degree of utilization of NASA Standard components, e.g., transponders, tape recorders and batteries, and deriving all science and engineering subsystems from current JPL spacecraft designs. The major exceptions to this are space-storable chemical propulsion and the spacecraft design chosen for the outer planets missions. It should be noted that engineering subsystem masses were not derived through a detailed design study, but were chosen as being representative of a design appropriate to meeting mission objectives.

In order to assure that the Ballistic Parity Option objectives can be achieved, it is necessary to assume that on-orbit assembly of upper stages and mission payload is a viable procedure during the launch opportunity time frame.

SEP-option mission payloads for the comet and asteroid missions are integrated with the SEP Stage, but are separable, free-flyer spacecraft for the planetary missions.

Deep-space and encounter chemical propulsion systems were optimally sized via the stage sizing rocket equation with the following parameter values:

	Solid	Monoprop	Earth-Storable	Space-Storable
Isp(sec)	285	215	290	370
Tankage factor	0.087	0.20	0.14	0.16
Engine mass (kg)	--	30	65	75

GENERAL TECHNICAL GUIDELINES AND ASSUMPTIONS

- 1985-2000 LAUNCH OPPORTUNITY TIME FRAME
- STATE-OF-ART HARDWARE TECHNOLOGIES
- JPL SPACECRAFT DESIGNS (e.g. Mariner, Voyager, Galileo)
- ON-ORBIT-ASSEMBLY OF UPPER STAGES IF NECESSARY
- PAYLOAD-SEPS INTERFACE
  - o INTEGRATED FOR COMET AND ASTEROID MISSION
  - o SEPARABLE FOR PLANETARY MISSIONS
- RUBBER SIZING OF SPACECRAFT DEEP-SPACE AND ENCOUNTER PROPULSION

For the purposes of this study, the following general guidelines and assumptions were used in developing the various cost estimates.

In keeping with current reporting practices, all cost estimates are presented in Fiscal Year 1982 constant dollars and 20% contingency is applied to the project cost portion of the total estimate in the form of Allowance for Program Adjustment/Management Reserve.

The recurring rates for the transportation-related cost elements were arrived at by mutual agreement with cognizant personnel at NASA, JPL and SAI. These rates are only representative of current best estimates, used for the purpose of cost trade-off analysis, and therefore should not be viewed as firm final estimates for these systems.

Each mission in the set of missions analyzed was treated independently, as if the project hardware was unique to that mission. No allowance was made in the costing for possible block buys of common hardware in, for example, the comet and main-belt asteroid missions or the two outer planet missions. However, heritage from prior projects or designs was taken into account, where appropriate, on a mission-by-mission basis.

Chemical propulsion systems were sized as required for each mission application. However, a standard set of assumptions regarding inheritance was consistently applied in each case. Relatively small solid motors are assumed to be commercially available in the appropriate size and are thus costed as off-the-shelf hardware. Design and fabrication of hydrazine systems is assumed to be well-understood technology, requiring minimal design effort concerning individual applications. Earth-storable bipropellant systems have been successfully flown in the past, but a considerable design effort is required for each application. Space-storable bipropellant systems have never been flown and thus would require a completely new development effort. For applications requiring more than one stage, the second and third stages are costed as exact repeats and/or block buys of the first stage.

Finally, the SAI Cost Model requires specific inputs concerning engineering and science subsystem masses and their associated heritage. For most missions analyzed, such masses were obtained from a variety of sources as deemed appropriate to the particular mission application. These sources included both existing designs and results of pre-Phase A design studies.

GENERAL COSTING GUIDELINES AND ASSUMPTIONS

- ALL COSTS PRESENTED IN FISCAL YEAR 1982 CONSTANT DOLLARS
- 20% CONTINGENCY (APA/RESERVE) APPLIED TO PROJECT-RELATED MISSION COST ELEMENTS
- TRANSPORTATION-RELATED FIXED RATE RECURRING COSTS

o Solar Electric Stage	\$ 70M/unit
o SEP Operations	\$300K/month
o Star 48	\$ 6M/unit
o IUS(II)	\$ 25M/unit
o Centaur (Wide Body)	\$ 50M/unit
o STS Operations	\$ 48M/launch
o On-Orbit-Assembly	\$ 25M/assembly

- NO ALLOWANCES MADE FOR POSSIBLE HARDWARE COMMONALITY/BLOCK BUY PHILOSOPHIES ACROSS THE MISSION SET ANALYZED
- CHEMICAL PROPULSION GENERAL INHERITANCE CLASSIFICATIONS

  o Solid retro motor : off-the-shelf

  o Hydrazine monopropellant : exact repeat

  o Earth-storable bipropellant : mix of exact repeat and minor modification

  o Space-storable bipropellant : essentially all new

- ENGINEERING SUBSYSTEMS DERIVED FROM VARIETY OF JPL SOURCES:

  o existing designs: Viking Orbiter, Voyager, Galileo

  o pre-Phase A design studies: LPO, HFTR, HIM, Outer Planets

The objective of the near-earth asteroid rendezvous mission is to reconnoiter several asteroids by rendezvous encounters. The first target of the mission is to be an asteroid in near-earth space in order to maximize the probability of mission success. Subsequent asteroids are only limited by performance and would likely be main-belt objectives due to better frequency of opportunity. Also, the science payload allowance for this mission is somewhat reduced at 80 kg.

The SEPS delivery option assumes a "bolt-on" science/communications payload of 300 kg. The near-earth asteroid is Eros, an S-Type object approximately 20 km in diameter. Two subsequent main-belt asteroids are included in the tour, i.e. Ilona (18 km diameter) and Drakonia (33 km diameter), to complete a three-asteroid flight profile. Rendezvous time (staytime) is 60 days at each object based on proposed science requirements. A Shuttle/IUS(II) launch of the SEPS-integrated payload occurs in December 1987 with the mission being completed 5.4 years later (including Drakonia staytime). A generous injected mass margin of 1048 kg exists permitting payload mass growth, a large launch window, or the possibility of some other asteroid sequence at a later launch date. Note that the 300 kg payload is not really a complete spacecraft but only consists of those subsystems needed to support the science instruments, which are not found on the SEPS. The SEPS is therefore retained throughout the mission, performing all station-keeping and navigation maneuvers as well as the inter-asteroid transfers.

The Ballistic Baseline delivery option assumes a complete free-flyer spacecraft of 551 kg. The science payload is the same as that assumed for the SEPS delivery option. A representative two-asteroid flight profile was found for this option within the launch capability of the Shuttle/Centaur (WB) vehicle. The near-earth asteroid is Anza, a small C-Type object approximately 3 km in diameter. The second target is the main-belt asteroid Nanetta (43 km diameter). The launch opportunity for this mission occurs in October 1988. Total mission time is 4.7 years including over a year's staytime at Anza due to celestial mechanics constraints. A two-stage space-storable propulsion system is needed to perform three deep-space impulse maneuvers plus navigation and station-keeping totalling 5.6 km/sec. Although the spacecraft includes a 5% mass contingency (a common study ground rule on all spacecraft designs) there is virtually no injected mass margin. This clearly demonstrates the difficulty of performing multi-asteroid missions in a ballistic flight mode, even with high energy upper stages like the wide-body (WB) Centaur.

The Ballistic Parity delivery option assumes the same free-flyer spacecraft as the Ballistic Baseline option. Its mission, however, includes three asteroid targets to complete parity with the SEPS option. The flight profile is the same as the Baseline option through the first two targets, Anza and Nanetta. It is then extended to a total trip time of 6.1 years to include the third target, the main-belt asteroid Caprera (39 km diameter). Staytime at both Caprera and Nanetta is two months, but remains at over a year at Anza. Adding the third target to an already difficult ballistic mission raises propulsion requirements dramatically. Three orbit-assembled Centaurs and a three-stage space storable deep space propulsion system are needed to meet the energy requirements with only a small mass margin. The post-injection impulse requirements of this mission are over 8 km/sec.

NEAR-EARTH ASTEROID RENDEZVOUS MISSION DEFINITION

	MISSION PARAMETERS	DELIVERY OPTIONS		
		SEPS	BALLISTIC BASELINE	BALLISTIC PARITY
FLIGHT PROFILE	FLIGHT MODE	MULTI-REV	MULTI-IMPULSE DIRECT	MULTI-IMPULSE DIRECT
	LAUNCH DATE	DEC 1987	OCT 1988	OCT 1988
	ENCOUNTER SEQUENCE (BODY/DATE)			
	FIRST RENDEZVOUS	EROS/JUN 1989	ANZA/MAR 1991	ANZA/MAR 1991
	SECOND RENDEZVOUS	ILONA/DEC 1990	NENETTA/FEB 1993	NENETTA/FEB 1993
	THIRD RENDEZVOUS	DRAKONIA/DEC 1992		CAPRERA/SEP 1994
	INTERPLANETARY TRIP TIME	5.2 Years	4.5 Years	5.9 Years
	STAY TIME	2 Months/asteroid	12.5 and 2 Months	12.5, 2 and 2 Months
PROP.	SHUTTLE UPPER STAGE(S)	IUS(II)	CENTAUR (WB)	3 CENTAURS(WB) W/OOA*
	PAYLOAD DEEP-SPACE PROPULSION SYSTEM(S)	NONE	SPACE-STORABLE	SPACE-STORABLE
	NUMBER OF PAYLOAD PROPULSION STAGES	NONE	TWO	THREE
MASS PERFORMANCE	SPACECRAFT DRY MASS	300 kg	551 kg	551 kg
	PROBE MASS	-	-	-
		-	-	-
	PROPULSION WET MASS	-	4094	12151
	SEPS SYSTEM MASS	1550	-	-
	SEPS PROPELLANT MASS	1502	-	-
	LAUNCH VEHICLE ADAPTER MASS	100	73	160
	TOTAL INJECTED MASS	3452	4718	12862
	LAUNCH CAPABILITY (@ C_3 in km^2/sec^2)	4500 (@0.0)	4725 (@47.0)	13100 (@47.0)
	LAUNCH MARGIN	1048	7	238

* OOA: On-orbit assembly of three wide-body Centaurs

A basic guideline for the Near Earth Asteroid Rendezvous missions is to make maximum use of residual hardware and existing designs in order to achieve an as-low-as-practical project cost. Possible sources for components which satisfy this requirement are listed on the facing page.

For the SEPS Baseline Option, this hardware plus necessary additions and interfaces, such as thermal control, cabling and mounting brackets, is assumed to be simply bolted to the solar electric stage in a bare-essentials packaging configuration. Thus, all guidance and navigation and regulated power is provided by the stage. Inheritance classifications used for costing purposes are presented for each subsystem, showing the percentage of subsystem mass allocated to each inheritance category. Most of the mass is allocated to the exact repeat and minor modification categories because of the stated guideline. "Residual hardware" does not necessarily imply off-the-shelf items, since refurbishment and some interface redesign may be needed. Similarly, the MMS data system would likely need interface and software modifications.

For both ballistic options, the same residual hardware/existing designs were used as basic components about which a complete spacecraft was developed. Necessary components to form complete attitude control, power and structure subsystems, for example, were derived from existing pre-Phase A design studies. Off-the-shelf components now include NASA Standard batteries and an inertial reference unit.

Cruise phase mission operations are kept at a minimum level in order to minimize the total project cost for the SEP Baseline, the mission-related hardware attached to the stage is monitored approximately once per week for status checks. For both ballistic options, the spacecraft is monitored approximately once every other day because no other operational support (i.e. SEPS) is provided to fly these missions. Encounter operations for all options are performed at normal operational levels.

NEAR-EARTH ASTEROID RENDEZVOUS COSTING CONSIDERATIONS

- Engineering subsystems and science for SEPS Baseline Option derived from maximum utilization of residual hardware/existing designs:

  - o Viking Orbiter high gain antenna
  - o Voyager X-Band amplifier
  - o Mariner actuators
  - o MMS data system
  - o Galileo NIMS

  Hardware is costed with the following distribution* of subsystems among inheritance classifications:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	7	-	83	10	-
Thermal, Cabling & Pyro	-	-	50	-	50
Att & Art Control	-	50	25	-	25
Communications	-	78	22	-	-
Command & Data Handling	50	30	-	-	20
Science Instruments	-	29	20	20	31

- Engineering subsystems and science for ballistic options derived by using same hardware as above and adding necessary components to form a complete spacecraft:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	2	-	69	5	24
Thermal, Cabling & Pyro	-	-	50	-	50
Propulsion	-	100	-	-	-
Att & Art Control	32	34	17	-	17
Communications	-	78	22	-	-
Command & Data Handling	50	30	-	-	20
Power	48	-	52	-	-
Science Instruments	-	31	30	21	18

- Minimum-level mission cruise operations: monitor systems approximately once per week for SEPS option, approximately once per 48 hours for ballistic options.

*percentage of subsystem mass

Cost estimates for the Near Earth Asteroid Rendezvous case are presented on the facing page.

Costs are shown separated at two major levels: project costs and transportation costs. Project-level costs are further separated into major elements. Program Management also includes costs associated with mission analysis and engineering. Science Development and Chemical Propulsion are only those costs pertaining to subsystem-level development, fabrication and testing. Hardware development includes the same costs for all other engineering subsystems and all costs pertaining to flight system-level design engineering, integration, testing and quality assurance, and costs associated with ground support equipment. Launch + 30 Days Operations includes all pre-launch and launch operations and ground software development costs. Mission Operations includes all cruise and encounter flight operations costs while Data Analysis includes costs of reduction, documentation and analysis of science and engineering mission data. Transportation-level costs include all recurring costs associated with the particular launch system required for each mission option.

For this case, the cost of science development for the ballistic options is slightly higher than that of the SEPS Baseline because of differences in the ranging altimeter experiment assumed with these delivery options. The cost of hardware development for the ballistic options is significantly higher than the SEPS Baseline hardware because of the need to develop a complete spacecraft.

Differences in mission operations costs reflect differences in both cruise operations procedures and mission durations.

NEAR-EARTH ASTEROID RENDEZVOUS COST ESTIMATES

	SEPS Baseline	Ballistic Baseline	Ballistic Parity
PROJECT COST*			
Program Management/MAE	$ 6.4M	$ 14.8M	$ 17.6M
Science Development	35.9	36.7	36.7
Hardware Development	46.3	106.6	111.5
Chemical Propulsion	--	57.9	95.0
Launch + 30^d Operations	6.2	14.8	17.8
Mission Operations	8.1	19.9	29.4
Data Analysis	3.6	8.9	13.2
APA/Reserve	21.3	51.9	64.2
Sub-total	$ 127.8	$ 311.5	$ 385.4
TRANSPORTATION COST*			
Solar Electric Propulsion Stage	70.0	--	--
Solar Electric Propulsion Operations	18.7	--	--
Star 48	--	--	--
IUS(II)	25.0	--	--
Centaur (Wide Body)	--	50.0	150.0
STS Operations	48.0	48.0	192.0
On-Orbit-Assembly	--	--	25.0
TOTAL COST*	$ 289.5M	$ 409.5M	$ 752.4M

* All costs in FY'82M Dollars

The purpose of the comet rendezvous mission is to perform the first spacecraft reconnaissance of a comet and, by using the rendezvous encounter mode, extend these early findings to an exploration level of investigation. A staytime of 12 months is envisioned to monitor the comet dynamics as it recedes from the Sun. The short-period comet chosen is Tempel-2 with arrival planned 50 days before its 1994 perihelion passage. Rendezvous continues out to approximately 3 AU providing observations of the comet during a four-fold reduction in solar radiation. A broad spectrum of science instruments is included in the 150 kg science payload allowance for the flight.

The SEPS delivery option assumes an "integrated" mission module of 639 kg. The SEPS provides housekeeping power, a stable-platform, and station-keeping ability to complete the necessary spacecraft functions. Launch occurs in June 1990 on a Shuttle/IUS(II) vehicle. A 3.6-year direct flight achieves rendezvous with the comet in January 1994. No spacecraft propulsion is needed due to the "integrated" design assumption, i.e. the SEPS is responsible for all post-injection maneuvers. The 200 kg launch margin is in addition to an already applied 5% mass contingency on the spacecraft mass. Upgrading the launch vehicle upper stage to a Centaur (WB) would, of course, provide considerably more launch margin if needed.

The Ballistic Baseline delivery option has a free-flying spacecraft payload of 753 kg (including again a 5% contingency). To this is added a large earth-storable deep-space propulsion system to perform 2.7 km/sec of impulses required for rendezvous and station-keeping. The total injected mass of 3210 kg requires the Shuttle/Centaur (WB) launch configuration to reach a C_3 of 73.0 km^2/sec^2. Launch occurs in July 1989, about one year earlier than for the SEPS option. With the same January 1994 arrival date the trip time is, therefore, a year longer at 4.5 years. Though the launch mass margin is larger at 434 kg, no backup upper stages of better capability exist, as was the case in the SEPS option.

The Ballistic Parity delivery option is similar to the Ballistic Baseline option in most respects, except that a higher energy interplanetary transfer is used to achieve parity in launch date with the SEPS option. Launch is now in August 1990 and flight time is back down to 3.5 years. The additional performance needed to fly this trajectory is achieved by going to a space-storable spacecraft propulsion system. Propulsion system mass is also increased by 800 kg due to a decrease in C_3 associated with the shorter trip time transfer (post-launch impulse requirements of 3.9 km/sec more than negate this savings). The Shuttle/Centaur (WB) launch vehicle is still required with launch mass margin now reduced to 263 kg.

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COMET TEMPEL-2 RENDEZVOUS MISSION DEFINITION

	MISSION PARAMETERS	DELIVERY OPTIONS		
		SEPS	BALLISTIC BASELINE	BALLISTIC PARITY
FLIGHT PROFILE	FLIGHT MODE	DIRECT	MULTI-IMPULSE DIRECT	MULTI-IMPULSE DIRECT
	LAUNCH DATE	JUN 1990	JUL 1989	AUG 1990
	ENCOUNTER SEQUENCE (BODY/DATE)			
	RENDEZVOUS (50 days before perihelion)	JAN 1994	JAN 1994	JAN 1994
	INTERPLANETARY TRIP TIME	3.6 Years	4.5 Years	3.5 Years
	STAY TIME	12 Months	12 Months	12 Months
PROP.	SHUTTLE UPPER STAGE(S)	IUS(II)	CENTAUR (WB)	CENTAUR (WB)
	PAYLOAD DEEP-SPACE PROPULSION SYSTEM(S)	NONE	EARTH-STORABLE	SPACE-STORABLE
	NUMBER OF PAYLOAD PROPULSION STAGES	NONE	ONE	ONE
MASS PERFORMANCE	SPACECRAFT MASS	639 kg	753 kg	753 kg
	PROBE MASS	-	-	-
	PROPULSION WET MASS	-	1970	2763
	SEPS SYSTEM MASS	1550	-	-
	SEPS PROPELLANT MASS	932	-	-
	LAUNCH VEHICLE ADAPTER MASS	100	53	61
	TOTAL INJECTED MASS	3221	2776	3577
	LAUNCH CAPABILITY (@ C_3 in km^2/sec^2)	3423 (@9.0)	3210 (@73.0)	3840 (@61.2)
	LAUNCH MARGIN	202	434	263

Subsystems for the Comet Temple 2 Rendezvous case were basically derived from recent designs for the Halley Intercept Mission. For the SEPS Baseline Option, mass estimates for structure, thermal control and power were adapted from the Halley Flyby/Temple 2 Rendezvous Mission Module since these particular designs are better suited to this type of mission.

Off-the-shelf components include NASA standard inertial reference unit, transponder and tape recorder, and the batteries for the ballistic options.

COMET TEMPEL-2 RENDEZVOUS COSTING CONSIDERATIONS

- Engineering subsystems for SEPS Baseline Option derived from the Halley Flyby/Tempel 2 Rendezvous Mission Module and from the Halley Intercept Mission with the following distribution* of subsystems among inheritance classifications:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	-	2	-	-	98
Thermal, Cabling & Pyro	-	18	7	-	75
Att & Art Control	32	39	3	-	26
Communications	11	25	46	-	18
Command & Data Handling	39	39	15	-	7
Power	-	39	-	48	13
Science Instruments	-	8	26	19	47

- Engineering subsystems for the ballistic options derived from Halley Intercept Mission:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	-	2	42	-	56
Thermal, Cabling & Pyro	-	-	100	-	
Att & Art Control	32	38	3	-	27
Communications	11	25	46	-	18
Command & Data Handling	39	39	15	-	7
Power	57	-	43	-	-
Science Instruments	-	8	26	19	47

* heritage values given in percent of subsystem mass

Cost estimates for the Comet Temple 2 Rendezvous case are presented on the facing page.

Hardware development costs for the ballistic options are higher than that of the SEPS Baseline, reflecting a somewhat heavier, more complex system and the added requirement to integrate the chemical propulsion stages. Even though the non-propulsive hardware is the same for both ballistic options, the Ballistic Parity hardware development cost is greater due to higher costs of system-level testing and integration associated with the more expensive space-storable propulsion development.

Project-level mission operations costs for the ballistic options are greater than that of the SEPS Baseline principally because of the more complex mission profile involving the chemical propulsion stages.

COMET TEMPEL-2 RENDEZVOUS COST ESTIMATES

	SEPS Baseline	Ballistic Baseline	Ballistic Parity
PROJECT COST*			
Program Management/MAE	$ 15.1M	$ 18.0M	$ 19.7M
Science Development	74.6	74.6	74.6
Hardware Development	123.8	140.8	146.4
Chemical Propulsion	--	25.1	44.3
Launch + 30^d Operations	15.1	18.2	19.9
Mission Operations	46.6	58.6	54.1
Data Analysis	20.9	26.3	24.2
APA/Reserve	59.2	72.3	76.7
Sub-total	$ 355.3	$ 434.0	$ 460.0
TRANSPORTATION COST*			
Solar Electric Propulsion Stage	70.0	--	--
Solar Electric Propulsion Operations	16.6	--	--
Star 48	--	--	--
IUS(II)	25.0	--	--
Centaur (Wide Body)	--	50.0	50.0
STS Operations	48.0	48.0	48.0
On-Orbit-Assembly	--	--	--
TOTAL COST*	$ 514.9M	$ 532.0M	$ 558.0M

* All costs in FY'82M Dollars

The objectives of the main-belt asteroid mission are the same as those of the near-earth asteroid mission; the difference is in execution. The main-belt mission is a full-up project designed to explore as many main-belt objects as possible. Target diversity and ample science payload (113 kg) are two important elements characterizing this mission. Though main-belt objects are more difficult to reach than near-earth targets, they are thought to be more representative of the original asteroid population, based on the theory that near-earth objects are immigrants of the main-belt, perturbed to their present orbits by collisions and planetary gravity perturbations.

The SEPS delivery option example shown below includes four main-belt asteroids. They include Harmonia (S-Type, 118 km diameter), Tanina (S-Type, 14 km), Fortuna (C-Type, 228 km), and Sappho (U-Type, 84 km). Variation of both Type and size are apparent. Launch is in October 1988, but other multi-asteroid missions occur on a frequent basis (probably monthly). The SEPS-based integrated mission module weighs 603 kg and is launched with a Shuttle/IUS(II) vehicle. No spacecraft propulsion is needed with the "integrated" design approach. A total trip time of almost 8 years includes two month staytime at each asteroid to map their surfaces and gravity fields. The launch margin is essentially zero meaning other mission examples may include only three targets or require upgrading the upper stage to a Centaur (WB).

The Ballistic Baseline delivery option includes only two main-belt asteroid encounters. To stay within the capability of a Shuttle/Centaur (WB) launch a Mars swingby is also needed. This narrows launch opportunities to only one every 25-26 months (the spacing in Mars opportunities) and further limits the choice of asteroids. The requirement for free-flight capability increases the spacecraft mass to 728 kg (includes 5% contingency). Added to this is a large two-stage earth-storable propulsion system to perform a total of 4.8 km/sec of post-launch impulse to effect the two rendezvous. Launch margin is a narrow 44 kg. Staytime at the first asteroid, Harmonia, is lengthened to 3.8 months due to optimum celestial phasing considerations. The second target, Tanina, is reach in 4.1 years after launch, about the same time as in the SEPS option.

The Ballistic Parity delivery option adds two more asteroid targets to the Baseline profile to achieve parity with the SEPS option. These asteroids are Amalthea (U-Type, 48 km diameter), and Hersilia (C-Type, 111 km). A Mars swingby is still used to relieve launch requirements but the total post-launch impulse has increased dramatically to 8.7 km/sec. Retaining parity in the spacecraft (728 kg), three space-storable retro stages are used to meet this impulse demand. The resulting payload configuration weighs 18,340 kg and requires three Centaurs (WB) and four Shuttle launches to reach a C_3 of 20 km^2/sec^2! This extreme solution to performance requirements of multi-asteroid missions aptly illustrates the suitability of the low-thrust SEPS option.

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MAIN-BELT ASTEROID RENDEZVOUS MISSION DEFINITION

	MISSION PARAMETERS	DELIVERY OPTIONS		
		SEPS	BALLISTIC BASELINE	BALLISTIC PARITY
FLIGHT PROFILE	FLIGHT MODE	MULTI-REV	MULTI-IMPULSE SWINGBY	MULTI-IMPULSE SWINGBY
	LAUNCH DATE	OCT 1983	JUL 1988	JUL 1988
	ENCOUNTER SEQUENCE (BODY/DATE)			
	PLANETARY SWINGBY	N/A	MARS/JUN 1989	MARS/JUN 1989
	FIRST RENDEZVOUS	HARMONIA/JUL 1990	HARMONIA/JAN 1991	HARMONIA/JAN 1991
	SECOND RENDEZVOUS	TANINA/MAY 1992	TANINA/JUL 1992	TANINA/JUL 1992
	THIRD RENDEZVOUS	FORTUNA/JUN 1994		AMALTHEA/JAN 1996
	FOURTH RENDEZVOUS	SAPPHO/JUL 1996		HERSILIA/APR 1998
	INTERPLANETARY TRIP TIME	7.9 Years	4.1 Years	9.9 Years
	STAY TIME	2 Months/asteroid	3.8 and 2 Months	3.8, 15.4, 10.9 & 2 Mos
PROP.	SHUTTLE UPPER STAGE(S)	IUS(II)	CENTAUR (WB)	3 CENTAURS(WB) W/OOA*
	PAYLOAD DEEP-SPACE PROPULSION SYSTEM(S)	NONE	EARTH-STORABLE	SPACE-STORABLE
	NUMBER OF PAYLOAD PROPULSION STAGES	NONE	TWO	THREE
MASS PERFORMANCE	SPACECRAFT MASS	603 kg	728 kg	728 kg
	PROBE MASS	-	-	-
	PROPULSION WET MASS	-	6181	17395
	SEPS SYSTEM MASS	1550	-	-
	SEPS PROPELLANT MASS	1596	-	-
	LAUNCH VEHICLE ADAPTER MASS	100	97	217
	TOTAL INJECTED MASS	3849	7006	18340
	LAUNCH CAPABILITY (@ C_3 in km^2/sec^2)	3865 (@5.0)	7050 (@19.2)	20000 (@20.0)
	LAUNCH MARGIN	16	44	1660

* OOA: On-orbit assembly of three wide-body Centaurs

The engineering subsystems for the Main-Belt Asteroid Rendezvous case are the same as those used for the Comet Temple 2 Rendezvous. The main differences between the two cases are that the science payloads are somewhat different and the chemical propulsion systems for the ballistic options are sized differently. These differences will impact the hardware development costs as described on the next page.

MAIN-BELT ASTEROID RENDEZVOUS COSTING CONSIDERATIONS

- Engineering subsystems for SEPS Baseline Option derived from Halley Flyby/Tempel 2 Mission Module and from Halley Intercept Mission with the following distribution* of subsystems among inheritance classifications:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	-	2	-	-	98
Thermal, Cabling & Pyro	-	18	7	-	75
Att & Art Control	32	39	3	-	26
Communications	11	25	46	-	18
Command & Data Handling	39	39	15	-	7
Power	-	39	-	48	13
Science Instruments	-	21	22	21	36

- Engineering subsystems for the ballistic options derived from Halley Intercept Mission:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	-	2	39	-	59
Thermal, Cabling & Pyro	-	-	100	-	-
Att & Art Control	32	38	3	-	27
Communications	11	25	46	-	18
Command & Data Handling	39	39	15	-	7
Power	57	-	43	-	-
Science Instruments	-	21	22	21	36

* heritage values given in percent of subsystem mass

Cost estimates for the Main-Belt Asteroid Rendezvous case are presented on the facing page.

As with the Comet Tempel 2 Rendezvous case, hardware development costs for the ballistic options are higher than that of the SEPS Baseline. The individual estimates for the asteroid case differ from the respective estimates of the comet case because of different integration costs associated with different science payloads and different chemical propulsion. Because of the large impulse (ΔV) requirements, the development costs of the chemical propulsion systems are a significant portion of the total engineering hardware development effort.

Costs of project-level mission operations for the SEPS Baseline and Ballistic Parity Options are high due to the relatively long mission times. Because the Ballistic Baseline Option can achieve only two of four targets, the associated mission operations costs is significantly lower.

MAIN-BELT ASTEROID RENDEZVOUS COST ESTIMATES

	SEPS Baseline	Ballistic Baseline	Ballistic Parity
PROJECT COST*			
Program Management/MAE	$ 13 9M	$ 19.2M	$ 23.8M
Science Development	61.1	61.1	61.1
Hardware Development	120.0	147.4	155.6
Chemical Propulsion	--	50.8	112.0
Launch + 30^d Operations	13.8	19.4	24.3
Mission Operations	58.8	36.0	89.5
Data Analysis	26.4	16.2	40.0
APA/Reserve	58.8	70.0	101.3
Sub-total	$ 352.8	$ 420.2	$ 607.6
TRANSPORTATION COST*			
Solar Electric Propulsion Stage	70.0	--	--
Solar Electric Propulsion Operations	20.0	--	--
Star 48	--	--	--
IUS(II)	25.0	--	--
Centaur (Wide Body)	--	50.0	150.0
STS Operations	48.0	48.0	192.0
On-Orbit-Assembly	--	--	25.0
TOTAL COST*	$ 523.8M	$ 518.2M	$ 974.6M

* All costs in FY'82M Dollars

The purpose of the Mercury Elliptical Orbiter Mission is to obtain initial global mapping of the Mercury surface, improve the measurements of the planet's gravity field and magnetospheric interaction with the solar wind, and perform appropriate solar measurements from the orbiter platform established at 0.31-0.47 AU. A science payload allowance of about 100 kg is assumed for these tasks. A 12-hour orbit is selected for this mission concept with an initially high periapse altitude of 1000 km. Solar perturbations will, however, provide substantial variations to this value over the orbit lifetime of the mission which is set at 6 months.

The <u>SEPS</u> delivery option uses a multi-revolution flight profile to reach Mercury without the benefit of planetary swingbys. Launch requires a Shuttle/IUS(II) configuration and opportunities occur at least once per year. The spacecraft is a free-flyer jettisoned from the SEPS shortly before Mercury approach. Orbit capture is achieved with a small solid-rocket motor and orbit sustenance is provided by a hydrazine monopropellant system. The total SEPS payload is 649 kg which includes a 5% mass contingency in the spacecraft mass of 523 kg. Launch margin is absorbed by the minimum trip time objective which is 1.7 years.

The <u>Ballistic Baseline</u> delivery option uses the favorable 1994 double Venus swingby flight mode to perform this mission with a Shuttle/IUS(II)/Star 48 launch vehicle configuration. Even so, the payload requires a substantial earth-storable propulsion system to navigate the multi-swingby trajectory and perform the orbit capture maneuver. The propulsion system wet mass is 2638 kg and performs a total impulse requirement of 3.4 km/sec. The spacecraft dry mass is increased to 553 kg to integrate this large retro system, but is otherwise unchanged from the SEPS option. Total trip time is increased to 2.4 years and launch margin is 302 kg. Because of the swingby constraints, however, launch margin cannot be traded for flight time with this option.

The <u>Ballistic Parity</u> delivery option also uses the multi-Venus-swingby flight mode but presents performance for a series of these mission opportunities (1980-2000) instead of just the best one. Launches occur with a frequency of one every 2-3 years. The number of Venus swingbys varies from opportunity to opportunity being as little as one or as many as four for any particular mission. To meet all launch requirements, a Shuttle/Centaur (WB) launch vehicle is needed. Trip times vary but can take as long as 3.4 years to reach Mercury. The mass performance example shown is for the rather poor 1989 opportunity which has two Venus swingbys. The spacecraft propulsion system mass is increased to 3011 kg to achieve a post-launch impulse requirement of 3.6 km/sec. The most significant energy requirement is the launch C_3 which is increased to 43.5 km^2/sec^2. Even so the Centaur (WB) upper stage provides a comfortable launch margin of 1255 kg.

MERCURY ELLIPTICAL ORBITER MISSION DEFINITION

	MISSION PARAMETERS	DELIVERY OPTIONS: SEPS	DELIVERY OPTIONS: BALLISTIC BASELINE	DELIVERY OPTIONS: BALLISTIC PARITY
FLIGHT PROFILE	FLIGHT MODE	MULTI-REV	DOUBLE VENUS SWINGBY	MULTI-VENUS SWINGBY
	LAUNCH DATE	AT LEAST ONCE/YEAR	JUL 1994	ONCE EVERY 2-3 YEARS
	ENCOUNTER SEQUENCE (BODY/DATE)			
	FIRST PLANETARY SWINGBY	N/A	VENUS/SEP 1995	ONE TO FOUR (VENUS SWINGBYS)
	SECOND PLANETARY SWINGBY	N/A	VENUS/MAY 1996	
	ORBIT INSERTION (1000 km X 12 hr ORBIT)	MERCURY/L + 1.7 Yrs	MERCURY/DEC 1996	MERCURY
	INTERPLANETARY TRIP TIME	1.7 Years	2.4 Years	UP TO 3.4 Years
	STAY TIME	6 Months	6 Months	6 Months
PROP.	SHUTTLE UPPER STAGE(S)	IUS(II)	IUS(II)/STAR 48	CENTAUR (WB)
	PAYLOAD DEEP-SPACE PROPULSION SYSTEM(S)	SOLID/MONO-PROP	EARTH-STORABLE	EARTH-STORABLE
	NUMBER OF PAYLOAD PROPULSION STAGES	TWO	ONE	ONE
MASS PERFORMANCE	SPACECRAFT MASS	523 kg	553 kg	553 kg
	PROBE MASS	-	-	-
	PROPULSION WET MASS	126	2638	3011
	SEPS SYSTEM MASS	1550	-	-
	SEPS PROPELLANT MASS	1561	-	-
	LAUNCH VEHICLE ADAPTER MASS	100	57	61
	TOTAL INJECTED MASS	3860	3248	3625
	LAUNCH CAPABILITY (@ C_3 in km^2/sec^2)	3860 (@5.0)	3550 (@19.4)	4880 (@43.5)
	LAUNCH MARGIN	0	302	1255

Subsystems for the Mercury Orbiter case were basically derived from pre-Phase A design studies for the Mercury Dual Orbiter and Lunar Polar Orbiter. Engineering subsystems for all three options include a hydrazine propulsion system for on-orbit maneuvers. Both ballistic options have added structure to accommodate the large orbit insertion propulsion stage.

The relatively small solid motor needed for orbit insertion for the SEPS Baseline Option is assumed to be commercially available and is therefore costed as an off-the-shelf item. Other off-the-shelf components include NASA standard transponder, tape recorder and batteries.

MERCURY ELLIPTICAL ORBITER COSTING CONSIDERATIONS

- Engineering subsystems derived from Mercury Dual Orbiter and from Lunar Polar Orbiter with the following distribution* of subsystems among inheritance classifications:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	-	-	58	-	42
Thermal, Cabling & Pyro	-	1	19	-	80
Propulsion	-	100	-	-	-
Att & Art Control	-	23	77	-	-
Communications	18	48	34	-	-
Command & Data Handling	33	-	67	-	-
Power	41	-	23	-	36
Science Instruments	-	70	-	20	10

- Solid Motor for SEPS Baseline Option costed as 100% Off-the-Shelf

* heritage values given in percent of subsystem mass

Cost estimates for the Mercury Orbiter case are presented on the facing page.

Hardware development costs for the ballistic options are somewhat higher than that of the SEPS Baseline Option because of the increased requirements for integrating the larger orbit insertion propulsion systems.

Project-level mission operations for the Ballistic Baseline show higher cost than the other options because of the longer mission time associated with a double Venus swingby. The operations cost shown for the Ballistic Parity Option was estimated assuming a single Venus swingby. Launching this option in less optimal opportunities would significantly increase the mission cost for this option.

MERCURY ELLIPTICAL ORBITER COST ESTIMATES

	SEPS Baseline	Ballistic Baseline	Ballistic Parity
PROJECT COST*			
Program Management/MAE	$ 9.4M	$ 12.0M	$ 12.5M
Science Development	38.7	38.7	38.7
Hardware Development	83.2	94.5	97.1
Chemical Propulsion	0.7	[illegible]	31.6
Launch + 30^d Operations	9.2	11.9	12.4
Mission Operations	16.4	22.1	15.2
Data Analysis	7.4	10.0	6.8
APA/Reserve	33.0	43.2	42.9
Sub-total	$ 198.0	$ 259.2	$ 257.2
TRANSPORTATION COST*			
Solar Electric Propulsion Stage	70.0	--	--
Solar Electric Propulsion Operations	5.7	--	--
Star 48	--	6.0	--
IUS(II)	25.0	25.0	--
Centaur (Wide Body)	--	--	50.0
STS Operations	48.0	48.0	48.0
On-Orbit-Assembly	--	--	--
TOTAL COST*	$ 346.7M	$ 338.2M	$ 355.2M

* All costs in FY'82M Dollars

The objective of the Outer Planet Dual Probe Flyby Mission is to deliver the first atmospheric probes to the planets Uranus and Neptune, and to conduct exploration-level science from the carrier spacecraft during its flyby. The probe designs are based on Galileo probe science capabilities. The mission concept is to have a single carrier deliver both probes to their respective targets using Uranus/Neptune swingby trajectories which exist throughout the decade 1985-1995. Carrier science payload allowance is set at 100 kg and should permit considerable enhancement over Voyager 2 instrument capabilities at these outer most planets. The entry probes are assumed to have the same penetration capability as the Galileo probe, i.e. to a depth of about 20 bars.

The SEPS delivery option uses a SEEGA flight mode to achieve sufficient energy to fly the Uranus/Neptune swingby trajectory. Launch opportunities for this mode exist once a year throughout the decade. A Shuttle/Centaur (WB) launch vehicle provides sufficient earth escape velocity so that the SEPS can deliver its payload to the fast swingby trajectory with a single SEEGA (rather than $SEEGA^2$). Trip time, including the initial SEEGA phase, to Uranus is 6.8 years, and to Neptune is 10.2 years. The low-thrust portion of the flight is over shortly after the earth swingby which concludes the SEEGA phase, at which point the SEPS is jettisoned. The carrier spacecraft is, therefore, a free-flyer and has a mass estimate of 725 kg. The total mass at SEPS jettison is 1448 kg which includes in addition to the carrier, two probes at 275 kg each and a hydrazine propulsion system for deep-space navigation of the swingby trajectory. The zero mass margin for launch simply means the trip time has been minimized. If payload growth were to exceed the built-in 5% contingencies assumed, the flight time would have to increase to absorb the performance short-fall. This characteristic is typical of all outer planet flight modes, whether they're low-thrust or ballistic.

The Ballistic Baseline delivery option uses a unique Jupiter gravity assist in combination with the reference Uranus/Neptune swingby trajectory to achieve a very fast three-planet flyby mission. Launch must occur in December 1992. The high departure C_3 (123 km^2/sec^2) is achieved with a Shuttle/Centaur (WB)/Star 48 configuration. The Jupiter swingby occurs less than two years after launch, Uranus is reached in 5.5 years, and Neptune's encounter in 8.2 years. The payload is identical to the SEPS option, since both spacecraft must be free-flyers, with the exception that the Baseline spacecraft carries somewhat more propellant to navigate through three encounters, instead of two. Total payload mass, therefore, increases to 1486 kg.

The Ballistic Parity delivery option uses a ΔVEGA (3+) flight mode to reach the high-energy Uranus/Neptune swingby trajectory, and thus achieves the same launch opportunity flexibility as the SEPS option, i.e. once/year. The ΔVEGA mode is not nearly as efficient as the SEEGA mode, however, even though both use the same launch vehicle. Flight time to Uranus is increased to 9.4 years, and total trip time to Neptune is almost 15 years. The payload is substantially increased by the requirement for a deep-space ΔVEGA maneuver. Using an earth-storable propulsion system with a wet weight of 3179 kg raises the total payload to 4454 kg.

OUTER PLANET DUAL PROBE FLYBY MISSION DEFINITION

	MISSION PARAMETERS	DELIVERY OPTIONS		
		SEPS	BALLISTIC BASELINE	BALLISTIC PARITY
FLIGHT PROFILE	FLIGHT MODE	SEEGA SWINGBY	MULTI-PLANET SWINGBY	ΔVEGA (3^+) SWINGBY
	LAUNCH DATE	ONCE/YEAR (1985-95)	DEC 1992	ONCE/YEAR (1985-95)
	ENCOUNTER SEQUENCE (BODY/DATE)			
	PLANETARY SWINGBY	EARTH/L + 2.2 Yrs	JUPITER/APR 1994	EARTH/L + 3.1 Yrs
	PLANETARY SWINGBY	URANUS/L + 6.8 Yrs	URANUS/MAY 1998	URANUS/L + 9.4 Yrs
	FLYBY	NEPTUNE/L + 10.2 Yrs	NEPTUNE/FEB 2001	NEPTUNE/L + 14.8 Yrs
	INTERPLANETARY TRIP TIME	10.2 Years	8.2 Years	14.8 Years
	STAY TIME	-	-	-
PROP.	SHUTTLE UPPER STAGE(S)	CENTAUR (WB)	CENTAUR (WB)/STAR 48	CENTAUR (WB)
	PAYLOAD DEEP-SPACE PROPULSION SYSTEM(S)	MONOPROP	MONOPROP	EARTH-STORABLE
	NUMBER OF PAYLOAD PROPULSION STAGES	ONE	ONE	ONE
MASS PERFORMANCE	SPACECRAFT MASS	725 kg	725 kg	725 kg
	PROBE MASS for URANUS	275	275	275
	for NEPTUNE	275	275	275
	PROPULSION WET MASS	173	211	3179
	SEPS SYSTEM MASS	1550	-	-
	SEPS PROPELLANT MASS	1555	-	-
	LAUNCH VEHICLE ADAPTER MASS	100	39	71
	TOTAL INJECTED MASS	4653	1486	4525
	LAUNCH CAPABILITY (@ C_3 in km^2/sec^2)	4653 (@47.0)	1486 (@123.0)	4550 (@50.0)
	LAUNCH MARGIN	0	0	25

Subsystems for the Uranus/Neptune Dual Probe Flyby spacecraft were derived directly from the innovative spacecraft design of the Outer Planets study. The goal of the design was a low-mass spacecraft using new component technology and therefore the overall design has relatively little heritage.

The atmospheric entry probes for both Uranus and Neptune are derived directly from the Galileo Probe with the major difference involving reduced heat shield mass.

The Ballistic Parity Option spacecraft has increased structure to accommodate integration of the chemical propulsion system used for the ΔVEGA deep space maneuver.

For the SEPS Baseline Option, SEPS Operations are costed under the assumption that the stage is jettisoned after 24 months of operation.

OUTER PLANET DUAL PROBE FLYBY COSTING CONSIDERATIONS

- Engineering subsystems and science derived directly from innovative low-mass low-heritage spacecraft design for Outer Planets Study with the following distribution* of subsystems among inheritance classifications:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	-	-	-	-	100
Thermal, Cabling & Pyro	-	-	40	-	60
Propulsion	-	100	-	-	-
Att & Art Control	-	-	-	-	100
Communications	-	-	5	-	95
Command & Data Handling	-	-	-	-	100
Power	-	-	50	25	25
Science Instruments	-	47	28	13	12

- Uranus and Neptune Probes derived from Galileo Probe; cost for both probes derived by concurrence with NASA/ARC.

- SEP Stage jettisoned after 24 months operation.

* heritage values given in percent of subsystem mass

Cost estimates for the Uranus/Neptune Dual Flyby case are presented on the facing page.

Costs of hardware development for the two baseline options are the same, while that for the Ballistic Parity Option is slightly higher, reflecting the cost of integration of the chemical propulsion stage. The cost estimate for probes development includes the costs of probe science, test and integration.

Costs shown for project-level mission operations directly reflect the long mission durations associated with missions to the far outer planets.

OUTER PLANET DUAL PROBE FLYBY COST ESTIMATES

	SEPS Baseline	Ballistic Baseline	Ballistic Parity
PROJECT COST*			
Program Management/MAE	$ 18.8M	$ 18.8M	$ 21.2M
Science Development	40.9	40.9	40.9
Hardware Development	225.0	225.0	230.2
Probes	117.6	117.6	117.6
Chemical Propulsion	--	--	30.0
Launch + 30^d Operations	19.0	19.0	21.5
Mission Operations	81.4	66.1	124.3
Data Analysis	15.7	12.7	23.7
APA/Reserve	103.7	100.0	121.9
Sub-Total	$ 622.1	$ 600.1	$ 731.3
TRANSPORTATION COST*			
Solar Electric Propulsion Stage	70.0	--	--
Solar Electric Propulsion Operations	7.2	--	--
Star 48	--	6.0	--
IUS(II)	--	--	--
Centaur (Wide Body)	50.0	50.0	50.0
STS Operations	48.0	48.0	48.0
On-Orbit-Assembly	--	--	--
TOTAL COST*	$ 797.3M	$ 704.1M	$ 829.3M

* All costs in FY'82M Dollars

The purpose of the Saturn Dual Probe Orbiter mission is to expand the exploration of the Saturnian system to a level comparable to that expected from the Galileo results at Jupiter. Two probes are included in this concept, however, to also address exploration of the Titan atmosphere. Both probes are assumed to be of the Galileo design, but the Titan probe has some expansion of capabilities for surface measurements. The Saturn probe would be deployed before orbit insertion, while the Titan probe would enter the satellite's atmosphere one orbit after insertion. The orbiter has a science payload allowance of about 140 kg and would conduct active investigations of the Saturnian system for two years after arrival.

The SEPS delivery option uses the SEEGA flight mode in combination with the Shuttle/Centaur (WB) launch vehicle to deliver the free-flyer payload to an Earth-Saturn transfer trajectory with a total flight time of 5.4 years. Launch opportunities occur once/year. An earth-storable propulsion system is included for navigation, Saturn orbit insertion, and orbital maneuvering. The SEPS payload of 4289 kg consists of the orbiter (761 kg), Saturn probe (300 kg), Titan probe (250 kg), and single-stage retro (2978 kg). A 5% mass contingency is included in the orbiter mass. The launch margin has been largely absorbed in flight time performance.

The Ballistic Baseline delivery option uses the unique 1998 Jupiter swingby opportunity to gain considerable flight time performance in reaching Saturn. Using the same Shuttle/Centaur (WB) launch configuration, Saturn can be reached in only 4.8 years. The payload configuration is the same, but the single-stage retro system is reduced in mass to only 1044 kg. The time lines at Saturn are unchanged from the SEPS-based mission profile. Launch mass margins are fully absorbed by flight time performance.

The Ballistic Parity delivery option must use a ΔVEGA flight mode to achieve the same launch opportunity flexibility as the SEPS option. Performance suffers considerably as a consequence. Using the same launch configuration of the Shuttle/Centaur (WB), the trip time to Saturn is increased to 7 years. The required spacecraft propulsion system is also increased to 3089 kg, even though all the payload inert masses are unchanged. This is, of course, the result of having to do two large deep-space impulse maneuvers, one for the ΔVEGA flight mode and one for orbit capture. The time lines are unchanged at Saturn and launch margins are again fully absorbed by flight time performance.

SATURN DUAL PROBE ORBITER MISSION DEFINITION

	MISSION PARAMETERS	DELIVERY OPTIONS		
		SEPS	BALLISTIC BASELINE	BALLISTIC PARITY
FLIGHT PROFILE	FLIGHT MODE	SEEGA	PLANET SWINGBY	ΔVEGA (3^+)
	LAUNCH DATE	ONCE/YEAR	MAY 1998	ONCE/YEAR
	ENCOUNTER SEQUENCE (BODY/DATE)			
	PLANETARY SWINGBY	EARTH/L + 2.2 Yrs	JUPITER/FEB 2000	EARTH/L + 3.1 Yrs
	PROBE RELEASE	SATURN/E - 5 Mos	SATURN/OCT 2002	SATURN/E - 5 Mos
	ORBIT INSERTION	SATURN/L + 5.4 Yrs	SATURN/FEB 2003	SATURN/L + 7.0 Yrs
	PROBE RELEASE	TITAN/E + 5 Mos	TITAN/JUL 2003	TITAN/E + 5 Mos
	INTERPLANETARY TRIP TIME	5.4 Years	4.8 Years	7.0 Years
	STAY TIME	24 Months	24 Months	24 Months
PROP.	SHUTTLE UPPER STAGE(S)	CENTAUR (WB)	CENTAUR (WB)	CENTAUR (WB)
	PAYLOAD DEEP-SPACE PROPULSION SYSTEM(S)	EARTH-STORABLE	EARTH-STORABLE	EARTH-STORABLE
	NUMBER OF PAYLOAD PROPULSION STAGES	ONE	ONE	ONE
MASS PERFORMANCE	SPACECRAFT MASS	761 kg	761 kg	761 kg
	PROBE MASS for SATURN	300	300	300
	for TITAN	250	250	250
	PROPULSION WET MASS	2978	1044	3089
	SEPS SYSTEM MASS	1550	-	-
	SEPS PROPELLANT MASS	1166	-	-
	LAUNCH VEHICLE ADAPTER MASS	100	49	70
	TOTAL INJECTED MASS	7105	2404	4470
	LAUNCH CAPABILITY (@ C_3 in km^2/sec^2)	7300 (@14.9)	2404 (@89.6)	4475 (@50.4)
	LAUNCH MARGIN	195	0	5

Subsystems for the Saturn Orbiter with Dual Probes spacecraft were derived directly from the innovative spacecraft design of the Outer Planets Study. The subsystems are essentially the same as those for the Uranus/Neptune Flyby case except that the hydrazine propulsion subsystem is not included in the two baseline options.

The atmospheric entry probe for Saturn was derived directly from the Galileo Probe with the major difference involving reduced heat shield mass. The Titan Probe is a new development, designed specifically for investigations at Titan.

For the SEPS Baseline Option, SEPS Operations are costed under the assumption that the stage is jettisoned after 33 months of operation.

SATURN DUAL PROBE ORBITER COSTING CONSIDERATIONS

- Engineering subsystems and science derived directly from innovative low-mass low-heritage spacecraft design for Outer Planets Study with the following distribution* of subsystems among inheritance classifications:

	Off-the-Shelf	Exact Repeat	Minor Mod	Major Mod	New
Structure & Devices	-	-	-	-	100
Thermal, Cabling & Pyro	-	-	40	-	60
Propulsion	-	100	-	-	-
Att & Art Control	-	-	-	-	100
Communications	-	-	5	-	95
Command & Data Handling	-	-	-	-	100
Power	-	-	50	25	25
Science Instruments	-	10	50	40	-

- Saturn Probe derived from Galileo Probe; Titan Probe is new; costs for both probes derived by concurrence with NASA/ARC.

- SEP Stage jettisoned after 33 months operation.

* heritage values given in percent of subsystem mass

Cost estimates for the Saturn Orbiter with Dual Probes case are presented on the facing page.

Costs of hardware development for all options reflect the costs of integration with the chemical propulsion stage. The cost estimate for probes development includes the costs of probe science, test and integration.

Costs shown for project-level mission operations directly reflect the nominal two year orbit mission.

SATURN DUAL PROBE ORBITER COST ESTIMATES

	SEPS Baseline	Ballistic Baseline	Ballistic Parity
PROJECT COST*			
Program Management/MAE	$ 22.8M	$ 22.0M	$ 22.8M
Science Development	58.2	58.2	58.2
Hardware Development	215.8	211.7	215.7
Probes	240.9	240.9	240 9
Chemical Propulsion	31.7	24.2	31.2
Launch + 30^{d} Operations	23.2	22.4	23.2
Mission Operations	106.7	98.8	120.1
Data Analysis	28.8	28.6	28.8
APA/Reserve	145.6	141.4	148.2
Sub-Total	$ 873.7	$ 848.2	$ 889.1
TRANSPORTATION COST*			
Solar Electric Propulsion Stage	70.0	--	--
Solar Electric Propulsion Operations	9.9	--	--
Star 48	--	--	--
IUS(II)	--	--	--
Centaur (Wide Body)	50.0	50.0	50.0
STS Operations	48.0	48.0	48.0
On-Orbit-Assembly	--	--	--
TOTAL COST*	$1051.6M	$ 946.2M	$ 987.1M

* All costs in FY'82M Dollars